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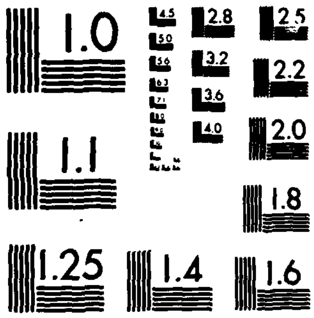

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Shuttle Detached Spacecraft

ROGER E. JACOBS

30 December 1983

AEROSPACE INSTRUMENTATION DIVISION PROJECT 7659
AIR FORCE GEOPHYSICS LABORATORY
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This report has been reviewed by the ESD Public Affairs Office (PA) and is releasable to the National Technical Information Service (NTIS).

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This report reports the results of a year-long in-house effort to investigate the potential utility and economic feasibility of developing a standard shuttle experiment support system to satisfy Air Force Geophysics Laboratory (AFGL) scientific requirements. It is shown that a shuttle detachable spacecraft capable of operating away from the orbiter environment, as well as attached in the cargo bay, maximizes its usefulness as an experiment support platform. A philosophy of minimum shuttle interface and efficient utilization of cargo bay volume is incorporated in an effort to reduce costs and provide greater flexibility for launch opportunities.

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Preface

Design of the Shuttle Detached Spacecraft (SDS) was influenced by AFGL/shuttle requirements and the author's concepts only. Detailed information about other free-flying systems was unknown to the author during the initial design phase of this study; therefore, any similarities which exist came about independently and unintentionally. The author would like to thank Edward McKenna for sparking the SDS concept, as well as Jack Griffin, Russell Steeves, and Christopher Krebs for helping with its development.

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Shuttle Detached Spacecraft

1. INTRODUCTION

The Shuttle Detached Spacecraft (SDS) is a new system concept designed to be a low-cost shuttle launched platform that will provide experiment operational support to many different AFGL scientific disciplines. Today, utilizing the shuttle as an experiment base is very attractive in many respects, however past experience has surfaced two major problems with its continued use. First, the basic ride cost is huge, made even greater by the complex interfaces required by todays state-of-the-art experiment systems; such as dedicated mission time, pointing, data storage, power, and so on. Secondly, the cargo bay has proven unsuitable for some of the more sensitive experiments due to pointing instabilities and lack of high order environmental cleanliness. We have addressed these concerns by developing a reusable experiment support platform concept which could operate with minimum orbiter interface both free-flying and attached in the cargo bay. It is designed to provide three-axis stabilized pointing, power, and onboard data processing/recording to its payload for up to five days to autonomous operation. All sub-systems make use of available technology and flight qualified components wherever possible to reduce development and certification costs.

(Received for publicati n 27 December 1983)

2. AFGL REQUIREMENTS

During the period of this study, there were eleven AFGL experiments submitted for space flight (DD1721) which could be considered candidates for an SDS-type shuttle launched support platform. Presently, AFGL experiments are integrated on a fixed pallet structure typically supplied by a shuttle integrating contractor; a process which has proven very expensive due to the huge amount of documentation, traveling, and other time-consuming interfaces needed to meet contractor requirements. SDS offers AFGL shuttle experimenters the basic convenience of having the entire system integrated and tested in-house where scientific needs and concerns could be handled more directly; therefore, more efficiently. Perhaps the greatest advantage SDS gives the scientist however, is the ability to operate autonomously from the shuttle as a free flying platform. This provides the versatility required by some experimenters not only to make measurements away from the contaminated shuttle environment, but to do so with high order pointing accuracies and station-keeping stability. SDS is a vehicle by which scientists could compare in- and out-of-bay data while, additionally, giving flight opportunities to those experiments previously excluded from the shuttle due to their hazardous operational nature. Positive feedback from laboratory scientists involved with plasma, particle, field, infrared, beam, and spacecraft charging experiments have attested to these needs and thus justified its conceptual development.

3. OVERALL DESIGN CHARACTERISTICS

AFGL experiments each have their own set of unique operating requirements, however, care was taken to not incorporate the very complex requirements of "high end" systems into the baseline design. Instead it was thought more advantageous to try meeting most experimenters' needs with as much operational simplicity as possible but, at the same time, have the ability to accommodate the "high end" user with only minimum modification. The structure was designed, therefore, to physically accept all sizes of lab experiments. By doing this, structural redesign is eliminated and the larger volume otherwise used for "sole occupant" class experiments would also allow sharing of the ride by smaller instruments thus further reducing per experiment costs.

Mixed NASA mission pricing policy defines a basic ride cost as the payloads' load factor multiplied by the total shuttle operational costs. The load factor being the payload's percent weight of 60,000 lb or percent length of 60 ft, whichever is greater. Therefore, to be most cost effective, a payload should occupy as little cargo bay length as possible but can weigh up to 1000 lb/ft without any cost penalty.

Additionally, regardless of the mission, the thinner the slice of bay needed by the payload, the more likely it will be given a ride on a "space available" basis. This implies that if you are going to break the 1000 lb/ft-guideline slightly it is better to be heavier than longer.

With thoughtful packaging it has been shown that the largest laboratory experiment, along with sub-systems enough to provide days* of autonomous support, could fit into a structure occupying as little as 5.5 ft of cargo bay length. This is made possible in part by incorporating the shuttle attach fittings right into the basic structural frame therefore eliminating the need for a cradling fixture which only adds weight and occupies valuable space. Sub-systems are designed for modular placement within the structure to facilitate easy modification, installation and repair.

4. SUB-SYSTEMS

To illustrate the following descriptions, please reference the sketch in Figure 1. Figure 5 includes a full summary of the proposed baseline sub-system characteristics.

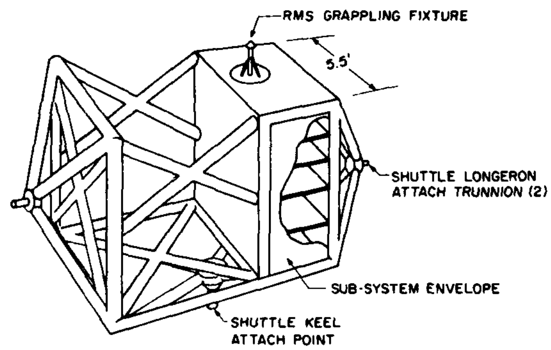

Figure 1. Shuttle Detached Spacecraft SDS

4.1 Structure

As previously mentioned, the SDS structure is sized to accommodate the largest of AFGL experiments, yet occupying only about 5.5 ft of cargo bay length. The structure has been sectioned for modular sub-system and experiment placement, the volumes of which were chosen to accommodate known hardware requirements and maximum consumable capacities. Figure 1 shows the scheme concept. Ideally, the structure should be of truss-like construction for strength, and have shuttle

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*Amount of on-orbit support time available to experiments depends on the experiments' particular power demands. See Section 4.

attach fittings as an integral part to conserve space and weight. This would result in the added flexibility of being able to use any of the over sixty deployable fitting locations for initial cargo bay placement and reberthing. Some candidate structural materials might be aluminum, fiberglass, or carbon fiber tubing, and should support a payload weight of approximately 6000 lb. Its component weight is estimated at 1500 lb.

4.2 Central Processor

SDS will incorporate a programmable central processing unit (CPU) that will handle command and control of all sub-systems. It will also provide the data processing function as well as sub-system monitoring and testing. Allocation of on-orbit time to various experiments, as well as contingency operational sequences, will be preprogrammed in the CPU.

4.3 Experiment

As previously noted, SDS was not designed for one particular experiment but is to provide general operational support to payload systems varying in size and power from 220 ft^3—3000 lb—2000 W (sole occupant class) to 1 ft^3—10lb—10W. Configuration of payload mounting has been left flexible to mission particular requirements; however, two possible standards are shown in Figures 2 and 3. Figure 2 depicts a sole occupant class infrared-red telescope, Figure 3 shows a "T" pallet mounting arrangement for multiple smaller payloads.

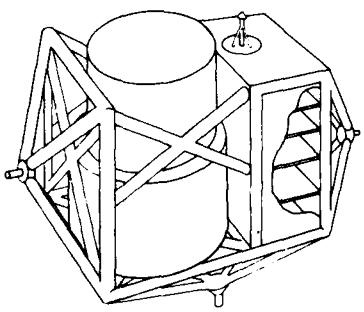


Figure 2. SDS-Large Sole Occupant Payload Configuration

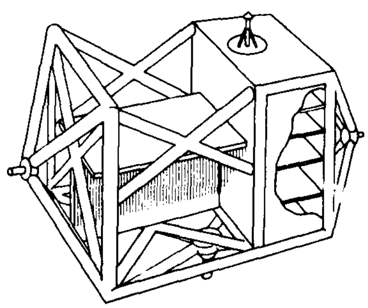


Figure 3. SDS-Multiple Payload Pallet Configuration

Power consumption in the experiment sub-system will have a direct effect on its continuous operational time as shown in Figure 4. Where the largest, most power demanding experiment might operate for about 30 hours, experiment operation of a multiple payload can be controlled so that the sub-system power draw with experiments operating either individually or together is low enough to return up to five days of data. This is important when considering the optimum data handling system for a particular mission.

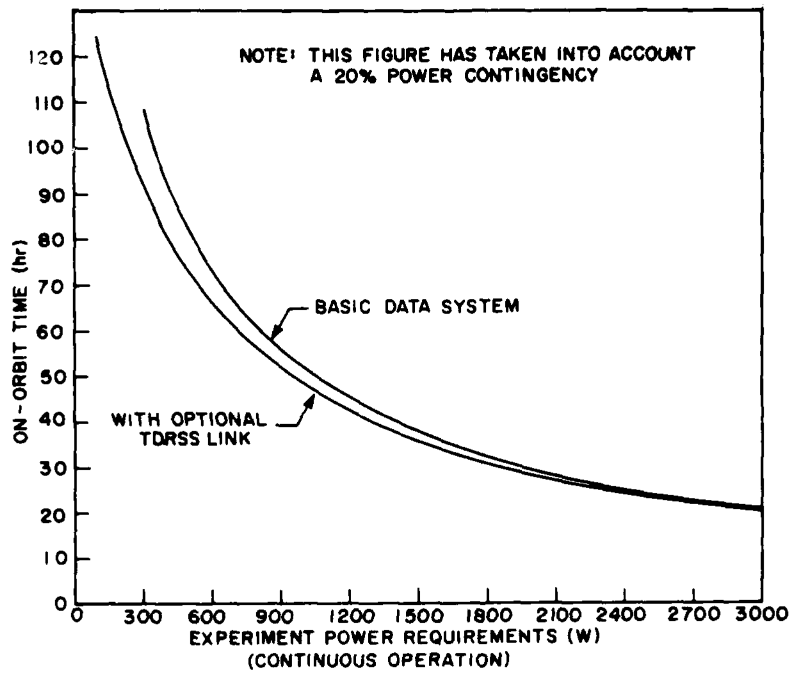


Figure 4. Total On-Orbit Time vs Experiment Power Requirements

4.4 Data

In an effort to keep SDS and its shuttle interface simple, data will be recorded on the spacecraft while free-flying, using the shuttle recorders when attached in the cargo bay. A good candidate for the on-board storage system is that used in CIRRUS 1A. Complete with electronics and sealed container, this system weighs 150 lb, uses 60 W of continuous power and records up to 32 hours providing 4×10^{10} bits of redundant storage. If a payload can be continuously supported beyond 32 hours

(see Figure 4), the following features of SDS make the extended on-orbit time fruitfull:

(1) The experiment is not required to and probably should not be taking data continuously. The central processor/sequencer can be programmed to run an experiment only at the most optimum times, that is, at a certain part of the orbit, looking at particular points of interest, after ACS gas dissipitation, and so on.

(2) The data processor function of the CPU can be used by the experiment to make more effective use of the available storage space by compressing and/or eliminating some of the raw data in the continuous stream.

It is feasible, therefore, that with data processing and non-continuous operation an on-orbit time of up to five days could be realized without exceeding the storage capacity of the recorders. If the experimenter still requires continuous raw data recording beyond 32 hours, however, or if the bit rates are just too high to be handled on-board, a TDRSS telemetry link could be installed as an option. When fully operational, TDRSS could be used for periodical data dumps either directly or through the orbiter to ground facilities at White Sands, New Mexico. In fact it is also feasible that AFGL could receive the data near-real-time through retransmission. In any case, all TDRSS scenarios would involve much greater expense due to hardware and interface costs.

4.5 Attitude Control

SDS will be three-axis stabilized in free flight by a cold gas attitude control system (ACS). The inertial platform will consist of rate gyros periodically updated by celestial sensor and/or tracker information resulting in a system pointing accuracy of approximately ± 3 arc min with less than 10 arc sec/min drift. This would meet all known AFGL pointing requirements. Cold N2 gas thrusting through coarse and vernier nozzles will provide both high and low rate rotational capability about each control axis. Advantage should be taken of the spacecrafts' size by positioning the nozzles at the dimensional extremities for most efficient gas utilization. There will be no translational capability built into the system and failure in this mode will be protected against by design.

4.6 Thermal Control

Thermal control will use passive devices such as paint, tape, and beta cloth whever possible. Analysis will show where and how in the structure certain subsystems should be mounted to best accommodate their needs and make best use of

their thermal characteristics. For the purpose of estimating the total system's power budget, however, it was assumed that some sort of active thermal control device would be required. Twenty-five (25) W of continuous power was allocated for this purpose as shown in Figure 5.

5. CONSUMABLES

The overall capabilities of SDS are dependant upon its ability to efficiently store and effectively use its consumables. The philosophy used in the preliminary design of consumable storage was to fill the assigned volumes to a realistic capacity with space approved, readily available components which are as light and cost efficient as possible.

5.1 ACS Gas

Chosen as candidate storage vessels are ten 1600 ft^3 filament wound bottles providing 140 lb of GN2 at 3000 psi. Other type vessels might have greater capacity, however those chosen are very lightweight, relatively inexpensive, and previously qualified for spaceflight on the shuttle. Total weight of bottles, gas and plumbing will be approximately 450 lb. At vehicle gross weight, calculations show this capacity should provide over 900 one-axis maneuvers.

5.2 Power

Silver zinc battery power has been chosen for our application mainly due to the advantage of rechargeability. Several packages are commercially available which include sealed containers with over-pressure valves to expel the hydrogen gas by-product overboard if it built up to excess. Generally, the amount of gas does not reach this point; however, any gas expulsion whatsoever would contaminate the environment and should be avoided. One manufacturer addresses this problem by providing nickel or silver hydrogen cells which can be placed within the battery container converting the hydrogen gas to water. In addition, their high capacity battery packs have previously flown on shuttle and therefore have been chosen as a good candidate for SDS. Each pack includes 23 cells in a flanged sealed container and provides 14 kWh of energy at a working 28 V. Six of these packs, along with the needed power distribution and monitor system, can be easily placed in the allotted volume, making the total sub-system energy capacity 84 kWh at a combined weight of approximately 2000 lb.

A maneuver being defined as movement from point to point at a rate of 1 deg/sec and stopping, thrust taking place at the structural extremeties of the spacecraft.

See Reference 1, page 23.)

System	Description	Compartment Dimensions	Weight (lb)	Power (Energy)	Operation Time
Structure	Truss-like, Single Bridge	15 ft × 15 ft × 5.5 ft	1500	---	---
Data Handling	4×10^{10} Bits Redundant Storage	15 ft^3	150	60 W	32 hr
TDRSS Link	Optional Sys. w/Transmitter	External Antenna	100	215 W	Intermittent
ACS	3-Axis Stabilized	26 ft^3	150	150 W	Continuous
ACS Gas	Filament Wound GN_2 Vessels	30 ft^3	450	---	---
Central Processor and Other Electronics	Programmable Self-Testing	22 ft^3	150	150 W	Continuous
Thermal Control	Active	---	---	25 W	Continuous
Batteries	Silver Zinc Cell Packs	22 ft^3	2000	84 kWh	Continuous
Experiment	Unknown	225 ft^3	3000 / 7500	Unknown	Unknown

Figure 5. SDS Sub-system Summary

6. SHUTTLE INTERFACES

6.1 Mechanical

SDS will make use of standard mechanical interfaces matching directly the active attach fittings and remote manipulator deployment sustem (RMS) of the orbiter. The hardware consists of a single bridge attachment in the bay (one longeron trunion on each side and one keel fitting) and a RMS grapple fixture conveniently placed on the top of the spacecraft (see Figure 1). There are over 60 active single bridge attach locations within the cargo bay available for use by deployables. An attempt should be made in the design to use any one of them for versatility.

6.2 Electrical

Presently there are two systems available for hardline electrical interface between a deployable spacecraft and the orbiter. One is a special purpose end effector (SPEE) connector which is on the RMS arm and mates and demates with RMS rigidization and spacecraft deployment. This interface can provide up to 35 A of 28 V power and 16 command/data lines, 8 of which are shielded. The second is an umbilical release, retract and retention system (SURS) previously flown and developed by Rockwell for the SPAS-01 pallet. It consists of power and command/ data umbilicals mounted in separate retraction/retention mechanisms on each side of the payload bay making connection to the spacecraft near its longeron trunnions. Each mechanism disconnects the umbilical electrically via command from the aft flight deck (AFD) then retracts and stows along the side of the cargo bay by spring tension. This system does not have the capability of remating on-orbit, though it does have EVA contingency for demating.

The mechanisms themselves are interchangeable by use of a common design, but the umbilicals are not (the power interface uses 19-12 AWG wires, where the command/data umbilical uses 100-22 AWG and 28-24 AWG shielded lines).

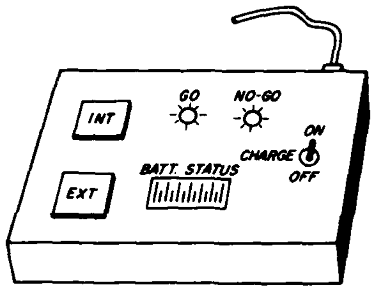
If in-bay SDS operation is required prior to RMS deployment, the SURS might be a good choice in order to conserve SDS onboard power and data storage space for the subsequent free-flight. If a pure free-flight was manifested, however, use of the SPEE connector would still provide the required initial turn-on and monitor functions while generally simplifying the overall shuttle interface. Mission particular requirements and further tradeoff analysis will dictate which, if not both, will be used. One possible hardline interface panel, along with its functional description is shown in Figure 6.

6.3 RF

The SDS system should also include an RF receiver for command interface with the orbiter. A simple hand-held transmitter in the Aft Flight Deck would suffice to command simple operational functions to the spacecraft. Although little work has been done to define this interface in any detail, ideas as to what might be included are shown in Figure 7. The interface as shown will be used later in describing nominal operational sequences.

6.4 Visual

Since SDS will not be capable of transmitting back to the orbiter, visual indications by exterior spacecraft lights will be used to indicate its status at certain times. This system includes GO/NO-GO status lights and strobe or self-illumination lighting for rendezvous. Description of when and how such lights might be used is included in the following operation sequence section and in Figure 7.



(COMMAND) (RESULT)

None - Battery charge circuit is always active.

Internal - SDS power bus is activated with internal battery power. RF receiver is on.

External - SDS power bus is activated with preconditioned orbiter power. RF receiver is on.

Charge - Supplies orbiter power to SDS where it's conditioned and used to charge internal batteries at a rate dependent on their condition.

CMP STATUS LIGHTS

GO/NO-GO - Illuminates according to results of RF test/standby command.

Figure 6. Control and Monitor Panel (CMP)

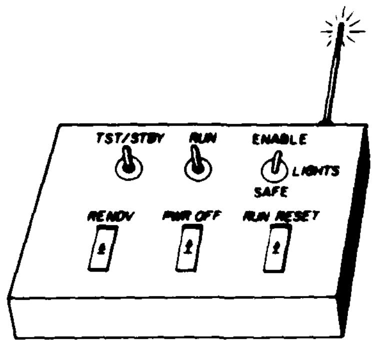
7. OPERATIONAL SEQUENCE

SDS will not be operated during launch or reentry, but for descriptive purposes, let us assume it will operate both attached and detached from the shuttle on orbit. At the prescribed point in the timeline, the SDS power bus will be energized by external orbiter power via CMP command. The mission specialist will then use the RF command box to send a TEST/STANDBY command which results in a "go" indication by status lights on both the CMP and the vehicle—the system is now in a standby mode. When a RUN command is given, SDS begins the preprogrammed "attached" operational routine. It distinguishes which program to run based on whether it is electrically connected to the orbiter or not. Upon completion of the program, the system returns to standby and its status is again indicated by lights on the CMP and on the spacecraft. At this time, the mission specialist switches to internal power in preparation for deployment and runs the self-test routine again. Once the system shows "go" for deployment, the umbilicals are released and the spacecraft is

deposited overboard by the RMS. With a subsequent RUN command, SDS initiates the preprogrammed "detached" routine sensing no attachment to the orbiter. When completed, the RENDEZVOUS mode is self-initiated to await pick-up. Due to differential drag between the orbiter and SDS, it is possible that separation distanced of up to 900 km can be experienced during a five-day period of time (see Figures 8-10). Once it's out of the bay, however, standard procedure is for ground-based C-band radar to track it, extrapolate its path when loss of signal occurs, and transmit its position upon request to the orbiter. The spacecraft's rendezvous lights will assist in visual identification by the orbiter crew for pick-up.

When the RMS has intercepted the vehicle, it should be commanded for a post-flight system test which subsequently places it in a standby mode for stowing in the bay. The status and rendezvous lights can be safed at any time. Once in the cargo bay, all spacecraft systems can be shut down by the POWER OFF command.

This was just one possible scenario to illustrate the nominal sequence of SDS operation and deployment events. If a SPEE interface is used, the scenario would be somewhat different but very similar. Certain contingency operational modes are included in the command functions as well; however, these scenarios will not be expounded upon at this time.



(COMMAND)		(RESULT)
Test/Standby	-	Tests all systems to be within predetermined limits. Routine terminates in mode where only essential systems are powered. Status lights are illuminated both on the CMP, if attached, and on the spacecraft as "go" or "no-go".
Run	-	Turns status lights off and runs "attached" or "detached" program depending on whether there is an electrical connection between SDS and the orbiter. If attached, the ACS will be safed, data will be fed to the orbiter recorders, and the program will terminate into a self-imposed test/standby mode. If detached, all systems are enabled for free flight and the program will terminate into a self-imposed rendezvous mode. The programs in this mode can be in interrupted without loss of sequence.
Reset	-	Initializes operating run program, rewinds tape and initiates a test/standby mode.
Rendezvous	-	Holds predetermined attitude with vernier thrusters and assumes minimum power operation awaiting pick-up by shuttle. Rendezvous lights are switched to the power bus for illumination if enabled.
Lights	-	Spacecraft light system can be enabled or safed by command.
Power Off	-	Internal power is disconnected from spacecraft power bus.

Figure 7. RF Command Box

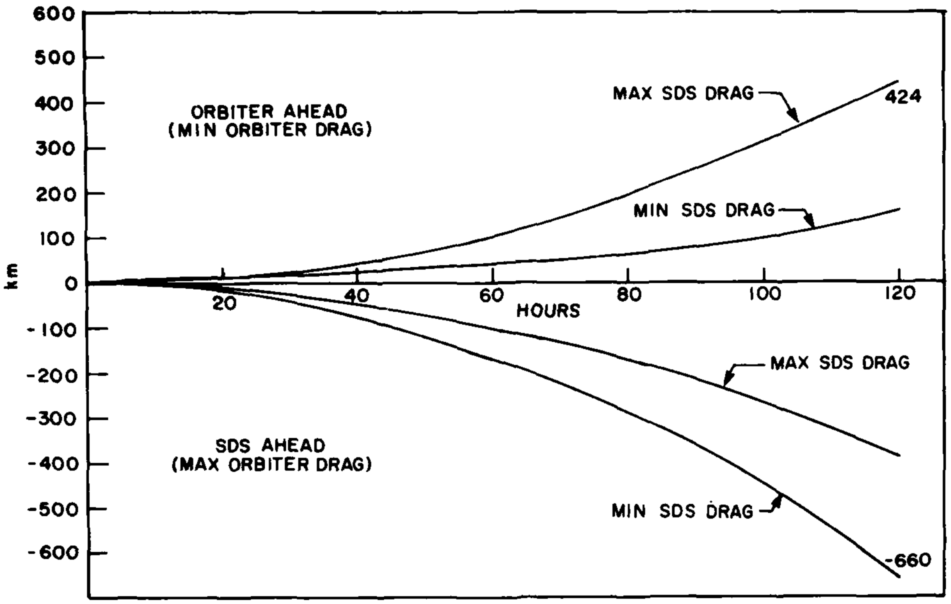


Figure 8. Separation Distance vs Time On-Orbit, 8000-lb SDS Weight

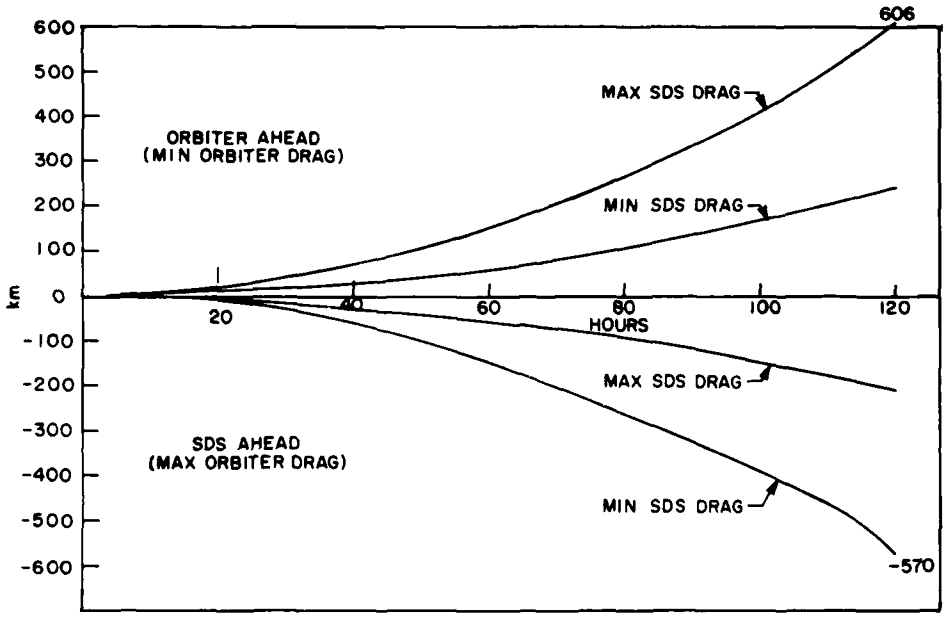


Figure 9. Separation Distance vs Time On-Orbit, 6000-lb SDS Weight

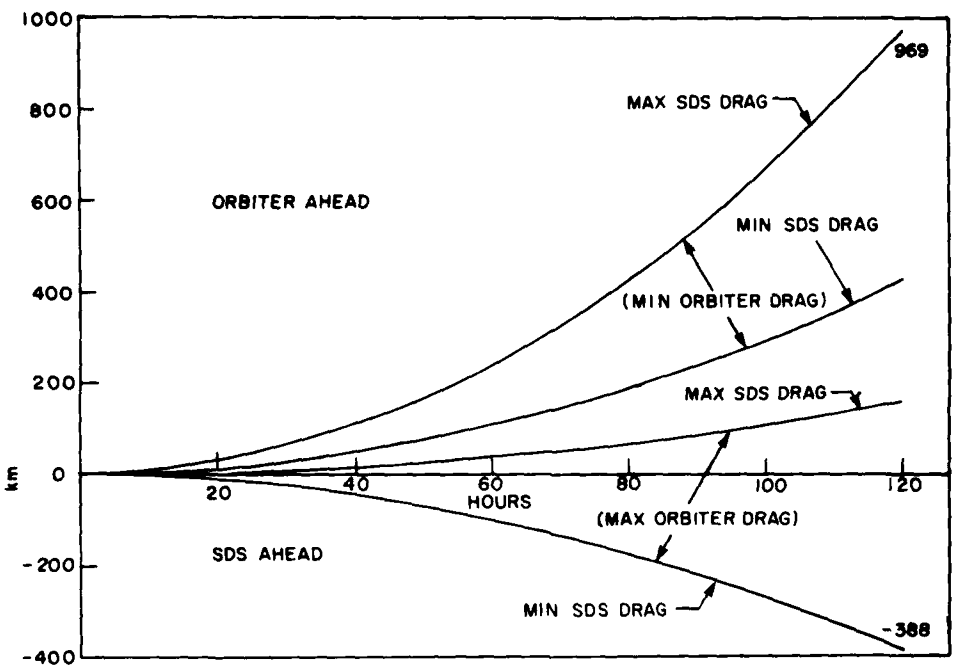


Figure 10. Separation Distance vs Time On-Orbit, 4000-lb SDS Weight

8. COST

The following cost sheet reflects a four-year development effort resulting in a fully operational and tested SDS experiment support system. The estimate includes refurbishment costs but does not reflect experiment, STS, NASA ground/radar support or shuttle integrator costs, if any.

Component	Initial Cost	Per Flight Refurbishment
ACS	$ 1.265 M	$0.460 M
Structure	0.435 M	0.060 M
Recorders	0.900 M	0.150 M
TDRSS (optional)	0.800 M	0.150 M
Central Processor	0.480 M	0.150 M
Power System	0.690 M	0.060 M
System Testing	0.500 M	0.500 M
Ground Support Equipment	0.500 M	-
System Integrator	3.275 M	2.000 M
Support Contracot	1.200 M	0.750 M
Subtotals		
Basic System	$ 9.245 M	$4.130 M
w/TDRSS option	10.045 M	4.280 M
Reserve		
Basic System	$ 0.925 M	$0.413 M
w/TDRSS option	1.005 M	0.428 M
TOTALS		
Basic System	$10.170 M	$4.545 M
w/TDRSS option	11.050 M	4.708 M

9. CONCLUSION

When comparing initial costs versus data return of SDS and sounding rockets, SDS is extremely impressive. That combined with its versatility and simplicity would facilitate widespread laboratory use and, therefore, should be considered as a natural progression into the shuttle era for AFGL experiment operations.

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